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HUGHES TOOL COMPANY · AIRCRAFT DIVISION

Culver City, California

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PERFORMANCE CALCULATION METHOD FOR A HOT CYCLE ROTOR

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for

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Figure 1. - Schematic of Hot Cycle Propulsion System

SECTION 1

SUMMARY

In accordance with Item 4e of Contract AF 33(600)-30271 (D/A Project No. 9-38-01-000, Subtask 616), which specifies submittal of a report describing performance calculations done under item 4b, equations are developed which permit computation of hovering rotor thrust for the hot cycle jet driven rotor used in the rotor whirl tests required by this Contract. These equations permit numerical evaluation of rotor performance on a digital computer, starting from gas generator discharge conditions and rotor geometry. The procedure employs classical thermodynamic relationships as extended by NASA to apply to a rotating blade, plus standard helicopter performance computation procedures. The overall system permits variation of the following parameters:

a. Hovering altitudes and temperature

b. Blade radius, chord, tip speed, and number of blades

c. Variation of blade profile drag coefficient

d. Duct cross-section area/blade cross-section area

e. Pressure drop in static duct between engine and blade root.

f. Duct friction coefficient

g. Tip nozzle velocity coefficient

h. Gas parameters: Mass flow, pressure ratio, temperature, gas properties

A numerical example is given for the computation of rotor power and thrust. The rotor geometry is for the 55 foot diameter rotor developed under the present contract. The gas conditions are those for two General Electric T64-6 turboshaft engines used as gas generators. The gas generators are operated at full power. It is assumed that water is injected into the tailpipe to limit the duct temperature to 900°F, the maximum permitted by the original duct material. The hovering condition is taken at 6000 feet - 95°F, out of ground effect. For this combination, the following results were obtained:

Blade duct inlet Mach number	M_3 = 0.39
Pressure ratio across blade	Tip/Root = 1.001
Rotor horsepower	1632
Rotor thrust	15,300 lbs.

Based on this calculation, the rotor was designed to a thrust of 15,300 lbs.

A later contractual modification authorized changing the duct material to the alloy, Rene' 41, which permitted use of full engine temperature of about 1200°F at full power. Under these conditions, the rotor would develop approximately 17,200 lbs thrust at 6000 feet, 95°F. To minimize cost, however, the rotor structure was not redesigned to the higher thrust level. Actually, the rotor, as built, permits operation at a load factor of 2.5 at a gross weight of 15,300. Operation at higher gross weight is permissible at proportionately reduced load factor limits.

SECTION 2

INTRODUCTION

This Contractor is required by Item 4e of Contract AF 33(600)-30271 (Reference 1) to submit a report describing the method of computing the thrust used to define the structure for the Hot Cycle rotor developed under that contract. This report describes the procedure for converting quantities such as gas generator discharge pressure,mass flow, and temperature associated with a hot cycle jet driven rotor into the more familiar form of horsepower available as found in the specification of any turboprop engine suitable for driving a helicopter rotor.

The general arrangement of the hot cycle propulsion system is shown schematically in Figure 1. A turbojet engine gas generator is used as a source of gas. All of the exhaust of the gas generator is led from the back of the engine up through the rotor pylon, past a rotating seal, to ducts in a multi-bladed rotor. The ducts end in aft-facing nozzles located at the tips of the blades. The blades are driven around by the jet reaction to the discharge of the gas. The rotor thus becomes a large diameter turbine, replacing the conventional small diameter turbine associated normally with free turbine engines. The exhaust from the jet engine is seen to drive the rotor around just as the rotor of a simple lawn sprinkler is driven by water supplied from a garden hose. The arrangement requires no gearing, leading to a very simple propulsion system.

The performance of a conventional gear driven rotor can be computed easily after the power available is obtained from the applicable engine specifications. This cannot be done with the configuration shown in Figure 1 because the specification for a gas generator will state only thrust, not power. The specification will include, however, the fundamental parameters of mass flow, pressure and temperature. These parameters can be converted to rotor power available, after making suitable allowance for duct friction and centrifugal pumping in the rotating blade. This conversion of parameters to rotor power available and then to rotor thrust is described below.

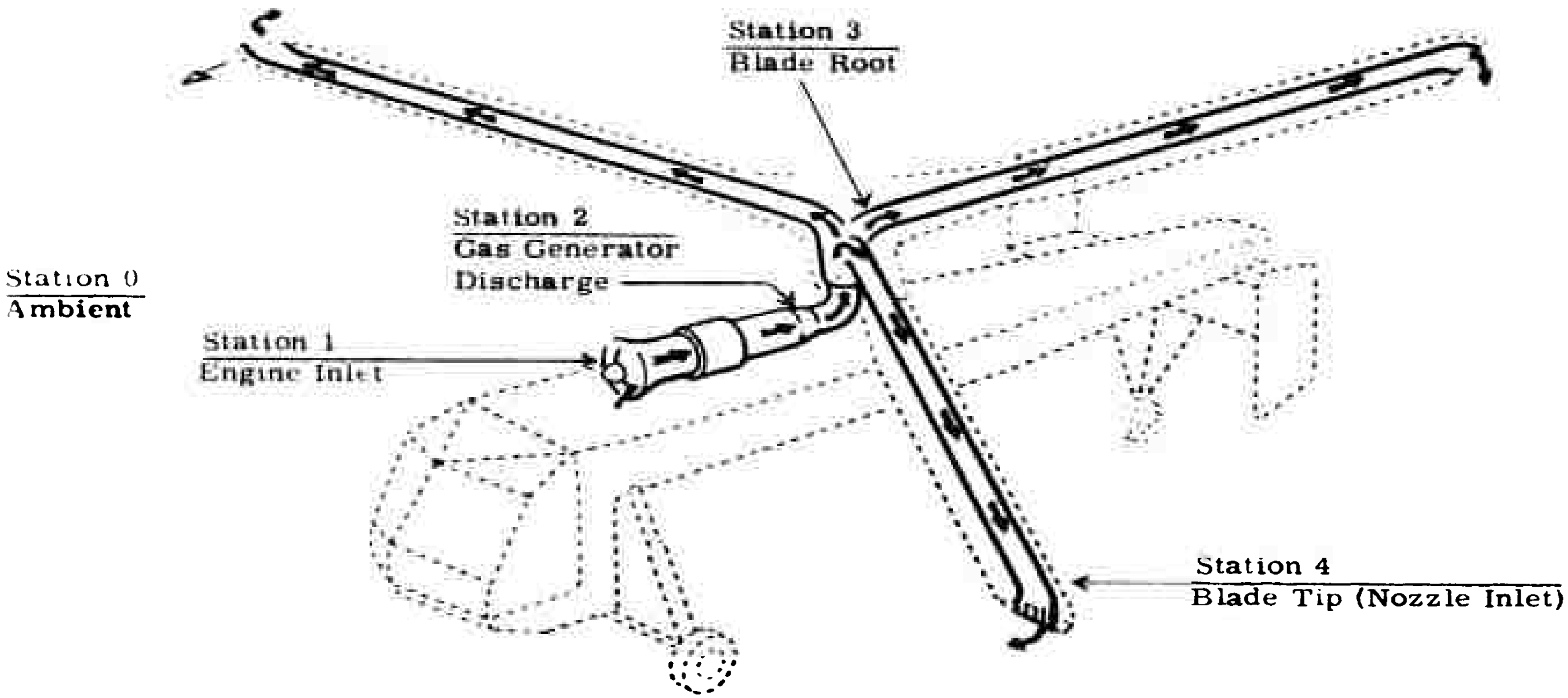

Figure 1. Hot Cycle Propulsion System

SECTION 3

SYMBOLS

A	Total duct cross-section area - sq ft
A_D	Duct cross-section area per blade - sq ft
a	Slope of lift curve - per radian
B	Tip loss factor (assumed = 0.97)
b	Number of blades
c	Blade chord - inches
C_i	Power correction factor for blade twist
c_p	Specific heat at constant pressure - BTU/lb - ^oF
C_P	Rotor power coefficient = RHP x $550/\rho \pi R^2 V_t^3$
C_T	Rotor thrust coefficient = $T/\rho \pi R^2 V_t^2$
c_v	Specific heat at constant pressure - BTU/lb - ^oF
C_{v_e}	Nozzle velocity coefficient
D	Duct hydraulic diameter = 4 x A_D/wetted perimeter
dr	Increment of blade length - feet
E	Number of gas generators
f	Duct friction coefficient
g	Acceleration of gravity, 32.2 ft/sec^2
J	Mechanical equivalent of heat, 778 ft-lb/BTU
M	Blade duct Mach number
n	Blade station index number. For 100 step computation, $n = 0, 1, 2, 3, \ldots\ldots 99$.

P_o	Ambient air pressure - lb/ft^2
P_T	Gas total pressure - lb/ft^2 (absolute)
R	Blade radius, feet Also, in equation (2) for flow function, = gas constant, 53.4 $\frac{ft\ lb}{lb - {}^oR}$
r	Gas total pressure ratio, P_T/P_o. Also, blade station, feet
(r_3/r_2)	Static duct total pressure ratio, $\frac{\text{blade root}}{\text{engine discharge}} = 0.96$
T	Rotor thrust, lbs
T_T	Gas total temperature - ^oR
(t/c)	Blade thickness ratio
$\frac{T_\infty}{T}$	Ground effect parameter $\frac{\text{thrust out of ground effect}}{\text{thrust in ground effect}}$
U	Duct utilization, $\frac{\text{duct area}}{\text{blade cross-section area}}$
V_j	Jet velocity at blade tip, ft/sec
V_t	Blade tip speed, ft/sec
W	Total gas flow, lb/sec = E x W_g
W_g	Gas flow per engine, lb/sec
γ	Ratio of specific heats, c_p/c_v
$\delta_0, \delta_1, \delta_2$	Coefficients in NASA profile drag polar
ρ	Air density at hovering altitude and temperature, $slug/ft^3$
σ	Blade solidity ratio = $\frac{b\ c}{12\pi R}$
Ω	Rotor angular velocity, radians/sec

<u>Subscripts</u> (See Figure 1)

0	Ambient air away from engine
1	Face of gas generator
2	Gas generator discharge
3	Blade root
4	Blade tip, at nozzle inlet

SECTION 4

COMPUTATION OF ROTOR POWER AVAILABLE

The pressure and temperature of the gas combine at the blade tip to produce a jet velocity V_j, which, together with the mass flow, W_g, from the engine, produce a gross force acting on the moving blade tip. As the gas moves from blade root to blade tip, the gas must be accelerated to the velocity of the blade tip, V_t, thus reducing the gross force to a net force. The net force itself moves forward at the blade speed, producing the net rotor power. Thus:

HP_A = Rotor Horsepower Available =

$$\frac{W_g}{g} \frac{(V_j - V_t)}{550} V_t \qquad (1)$$

This equation is very similar to that for the thrust hp of a turbojet engine, in which forward speed replaces tip speed of equation (1).

This expression can be broken down into terms,

$$\left(\frac{W_g V_j}{g \times 550}\right) \times V_t \quad \text{and} \quad \left(-\frac{W_g V_t}{g \times 550}\right) \times V_t$$

The first term is seen to be the gross thrust on the rotor blades, multiplied by the blade tip speed. This term is thus the gross power produced at the rotor blade tips by the gas exhaust. The second term is the force required to accelerate the gas to the blade tip speed, multiplied by the blade tip speed. This term is thus the power required to accelerate the gas to blade tip speed, and it is sometimes called the "pumping power" required to accelerate the gas. In a turbojet installation, an identical expression applies for the thrust power required to overcome ram drag.

Since tip speed and mass flow are known, the problem of determining rotor power available is reduced to finding the blade tip jet velocity, V_j.

4.1 Factors which Influence Jet Velocity, V_j

Because the jet velocity is a measure of the condition of the gas as it leaves the blade, it can be seen that various factors in the system will act to change the jet velocity from the value which would be

obtained if the gas at the back of the engine were exhausted directly to the atmosphere. The discussion given here will therefore describe the changes in the gas as it proceeds from the engine to the nozzle, substantiating the change when necessary

4.2 Gas Conditions at Discharge From Gas Generator

The gas conditions are always referred to the discharge from the gas generator turbine. Different engine manufacturers call this station by different designations, such as Station 5.1 by General Electric, or Station 7 by Pratt & Whitney. For purpose of this analysis the gas discharge station will be called Station 2, as shown on Figure 1. At that location the following are obtained.

Mass flow	W_g	lb/second
Total temperature	T_{T_2}	^oR
Pressure ratio	r_2	

4.3 Gas Conditions at Discharge From Static Duct

In this report, the phrase "static duct" is taken as the length of duct from the gas generator discharge, Station 2, through the rotor pylon rotating seal, to the rotor blade root, shown as Station 3 on Figure 1. This static duct thus includes a length of straight duct, a 90° turn, a straight duct, and a second 90° turn, ending just as the blade ducting starts. The portion of the static ducting above the rotating seal is seen in Figure 1 to transform from a single round duct at the rotating seal to three smaller round ducts leading to Station 3. The total pressure drop in this complex piece of ducting was measured as reported in Reference 2, and found to be less than 1% of the initial total pressure.

The rest of the static ducting below the rotating seal is more conventional and its pressure drop was conservatively estimated to be 2.0%. A diverter valve will also be in the ducting which will have a pressure drop of about 1%. Therefore the total pressure drop in the static duct is assumed to be 1% + 2% + 1% or 4% of the initial total pressure.

It is also assumed that no leakage will occur in the static duct, and that the duct itself is so short and effectively insulated by shrouding that no temperature drop will occur in the static duct.

The gas conditions leaving the static duct thus become:

Mass flow: $W_{g_3} = W_{g_2} = W_g$

Temperature $T_{T_3} = T_{T_2}$

Pressure ratio $r_3 = \frac{r_3}{r_2} \times r_2 = (1 - .04)\ r_2 = .96\ r_2$

4.4 Blade Duct Inlet Mach Number

The blade duct inlet Mach Number M_3 is the starting point to determine changes of properties of the gas as it flows from root to tip. This Mach Number M_3 is found from the following relationship, which is equation 4.16 of Reference 3 in consistent engineering units.

$$\left(\frac{W\sqrt{T_T}}{A\ \ P_T}\right)_3 = \frac{M_3\sqrt{\frac{\gamma g}{R}}}{\left(1 + \frac{\gamma - 1}{2} M_3{}^2\right)^{\frac{\gamma + 1}{2(\gamma - 1)}}} \qquad (2)$$

When all input quantities and constants are substituted in equation (2), the equation can be solved for M_3 by reading a graph of equation (2) or by iteration using a digital computer. The change in gas properties across the blade will now be found.

4.5 Change in Mach Number Through Blade Duct

The key to determination of the blade tip nozzle pressure ratio and temperature, which determine the jet velocity term V_j highlighted in equation 1, is to find the duct Mach number at the blade tip, Station 4. The relationship between tip Mach number, M_4, and root Mach Number, M_3, and the root pressure ratio, r_3, will determine the tip nozzle pressure ratio, r_4.

The subject of the behavior of a compressible fluid in the duct of a rotating jet helicopter blade is developed in Reference 4. That reference, which is an NACA Technical Note, considers all of the pertient factors of friction, centrifugal force, duct area change, and heat transfer. The reader is urged to review that report, which discusses the subject more thoroughly than can be done here.

Reference 4 will be seen to be a particular application of the more general problem of the flow of a compressible fluid subject to friction, heat transfer, and area change. Reference 4 states that the basis for study of the helicopter blade is the classical thermodynamic work of Shapiro given in Reference 3, (equation 8.40 of Chapter 8) plus the inclusion of centrifugal force turn, which was not considered in Reference 3. If the reader is not already familiar with the fundamental relationships developed by Shapiro, he is urged to study Chapter 8 of Reference 3 as well as Reference 4.

The basis for computation of change of Mach Number through the blade duct is Equation (2) of Reference 4. That equation will be evaluated numerically by a step-by-step procedure. It is therefore necessary to break the blade up into many small segments and compute the incremental Mach Number change across each segment. The tip Mach Number will then be the root Mach number plus the sum of the changes across all the segments.

Note 1: Constant Blade Duct Area

Equation 2 of Reference 4 includes a term dA/A which permits calculation of the influence of change of blade duct area. The hot cycle rotor built for this contract has ducts with the same area throughout the blade duct. Therefore dA/A is zero, and this term does not apply.

Note 2: Constant Gas Temperature From Root to Tip

Equation 2 of Reference 4 includes a term dT_T/T_T which permits calculation of the influence of heat transfer, or the change of total gas temperature. The hot cycle blade has a double wall type of construction which effectively insulates the gas stream. In addition, the work of compression done by the blade on the gas as it flows along the blade will add about 30°R to the gas temperature at the tip relative to the root. Some heat transfer will occur in spite of the insulating air gap. Preliminary calculations were made (which later whirl tests substantiated) that indicate that the net result of heat transfer and centrifugal compression is that the gas temperature is essentially constant from root to tip. That is, $T_{T_3} = T_{T_4}$. Therefor dT_T/T_T is zero, and this term does not apply.

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After eliminating the dT_T/T_T and dA/A terms from Equation 2 of Reference 4, as permitted by Notes 1 and 2 here, the equation for Mach number change becomes:

$$dM = \frac{2\gamma f M^3\left(1+\frac{\gamma-1}{2}M^2\right)dr}{D\left(1-M^2\right)} - \frac{M\left(1+\frac{\gamma-1}{2}M^2\right)^2 \Omega^2 r\,dr}{g\left(c_p - c_v\right)J T_T\left(1-M^2\right)}$$

Multiply first term on right side by $\frac{R}{R}$ and second term by $\frac{R^2}{R^2}$ and

substitute: $V_t^{\,2} = (\Omega R)^2$ (Blade tip speed)^2

$\Delta r = dr$ For finite small step instead of a differential

$\Delta M = dM$

$T_T = T_{T_3} = T_{T_2}$

$\left(\frac{2n+1}{2}\right)\left(\frac{\Delta r}{R}\right) = \frac{r}{R}$ (For a 100 step calculation $n = 0, 1, 2, \ldots\ldots\ldots, 99.$)

Note: The last substitution comes from the need to identify properly each segment in a calculation. For instance, assume a blade is broken in 10 steps from radius $r = 0$ to $r = R$. Therefore, 10 successive computations will cover the blade. Each step should be referred to the midpoint of a segment.

	n	0	1	2	3	4	5	6	7	8	9
r/R:	0	.1	.2	.3	.4	.5	.6	.7	.8	.9	1.0

—— $\frac{r}{R} = .35$ ——

Example:
Let $\frac{\Delta r}{R} = .1$ and $\frac{r}{R} = .35$

Then $n = 3$ and

$\left(\frac{2n+1}{2}\right)\left(\frac{\Delta r}{R}\right) = \frac{(2\times 3)+1}{2} \times .1 = .35$, which checks

Therefore:

$$\Delta M = \left(\frac{2f\gamma R}{D}\right)\left(\frac{\Delta r}{R}\right)\frac{M^3\left(1+\frac{\gamma-1}{2}M^2\right)}{(1-M^2)} - \frac{V_T^2 M\left(1+\frac{\gamma-1}{2}M^2\right)^2}{g(c_p-c_v)JT_{T_2}(1-M^2)} \times \left(\frac{2n+1}{2}\right)\left(\frac{\Delta r}{R}\right)^2$$

Where: D = duct hydraulic diameter, $\frac{4 \times \text{duct cross-sectional area}}{\text{Perimeter}}$

If D is known for the duct, it should be used. If it is not, a good approximation for two elliptical ducts in a NACA symmetrical airfoil is given by the following; taken from Appendix II, of Reference 5:

$$D = \frac{2.72\,(t/c)\ U \times c}{2\pi\sqrt{\frac{1}{8}\left(\frac{t}{c}\right)^2 + .0935\ U^2 + 1.75\frac{t}{c}}} \qquad (4)$$

The step by step computation using equation (3) starts with $M = M_3$ and $n = 0$, and ΔM is found. The computation proceeds until the last value of is used and the final Mach number is M_4

That is,

$$M_{(n=1)} = M_3 + \Delta M_{(n=0)}$$

$$M_{(n=2)} = M_{(n=1)} + \Delta M_{(n=1)}$$

$$M_4 = M_{(TIP)} = M_{(n=99)} + \Delta M_{(n=99)}$$

4.6 Pressure Ratio Produced By The Rotating Blade

The blade tip Mach number M_4 and the blade root Mach number M_3 can now be used to compute the pressure ratio across the blade, $\frac{r_4}{r_3}$. This value is given by equation (6) of Reference 4, as modified to account for constant duct area and constant gas temperature per Notes 1 and 2 of the last Section.

$$\frac{\text{Pressure at tip}}{\text{Pressure at root}} = \frac{r_4}{r_3} = \frac{M_3}{M_4}\left(\frac{1+\frac{\gamma-1}{2}M_4^{\,2}}{1+\frac{\gamma-1}{2}M_3^{\,2}}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \qquad (5)$$

The nature of this equation is such that if M4 $\langle$ M3, $\frac{r_4}{r_3} \rangle$ 1.0 and if $M_4 \langle M_3$, $\frac{r_4}{r_3} \rangle 1.0$. In the typical case, it has been found that M_4 is usually very nearly equal to M_3. As a result, r_4 will generally be found to be nearly equal to r_3.

4.7 Gas Conditions At Blade Tip Nozzle

It is now possible to determine the gas conditions at the blade tip nozzle. Because it was assumed that there is no leakage anywhere in the system, the mass flow at the nozzle is equal to that at the engine. The gas temperature at the tip is assumed to be the same as at the engine, because of the short static ducting, plus the cancelling of heat losses through the blade and temperature rise due to centrifugal pumping. Finally the nozzle pressure ratio r_4 can be determined from the preceding section.

Mass flow $\quad W_{g_4} = W_{g_3} = W_{g_2} = W_g$

Temperature $\quad T_{T_4} = T_{T_3} = T_{T_2}$

Pressure ratio $\quad r_4 = r_2 \times \frac{r_3}{r_2} \times \frac{r_4}{r_3} = r_2 \times .96 \times \frac{r_4}{r_3}$

4.8 Calculation of Jet Velocity and Rotor Power Available

The jet velocity V_j at the tip of the rotor blade is found from

$$V_j = C_{v_e}\sqrt{2\ g\ J\ c_p\ T_{T_4}\left[1 - \left(\frac{1}{r_4}\right)^{\frac{\gamma-1}{\gamma}}\right]} \qquad (6)$$

Equation (6) is derived from Chapter 4 of Reference 3 by rearranging Equation 4.4, 4.14a, and 4.14b.

For a typical high compression gas generator, the nozzle pressure ratio r_4 will be approximately 2.8, and T_4 will be about 1600^oR. With a velocity coefficient of 0.97, which can easily be obtained at the stated value of r_4, the jet velocity V_j will be about 2000 - 2100 feet second.

The rotor horsepower per pound per second of gas is found by rearranging Equation (1) as follows:

$$\frac{RHP}{W_g} = \frac{\left(V_j - V_t\right) \ V_t}{g \times 550}$$

For a jet velocity of 2100 feet/second and an average tip speed V_t of 700 feet per second, the $\frac{RHP}{W_g}$ will be

$$\frac{RHP}{W_g} = \frac{(2100 - 700) \ 700}{32.2 \times 550} = 55.3$$

Finally, the rotor horse power available is:

$$RHP = \frac{E \times W_g \times (V_j - V_t) \ V_t}{32.2 \times 550} \qquad (7)$$

This power available will now be used to determine the rotor thrust.

SECTION 5

COMPUTATION OF ROTOR THRUST

The computed thrust of the rotor in hovering is determined from the rotor power available and the rotor geometry using standard NACA methods presented in Reference 6. The approach used is to make the power available equal to the power required; the power required is the sum of the induced power and the profile power. If ground effect is present, its influence on induced power is determined from the empirical data presented in Figure 5-13 of Reference 6. The profile power coefficient presented in Reference 6 was obtained for a 12% thick blade. As the Hot Cycle blades are 18% thick, the profile power has been increased by 17% to allow for the increased drag of the thicker section. The blades are twisted -8^{o}; therefore the correction to the overall power coefficient presented on page 85 of Reference 6 is used. Applying the above factors, the equation for power as a function of thrust - Equation 36 of Reference 6 becomes:

$$C_P = \left[\frac{C_T^{3/2}}{\sqrt{2}\,B} \left(\frac{T_{\infty}}{T} \right)^{3/2} + 1.17 \left(\frac{\sigma \delta_0}{8} + \frac{2}{3} \frac{\delta_1}{a} \frac{C_T}{B^2} + \frac{4 \delta_2}{\sigma a^2} \left(\frac{C_T}{B^2} \right)^2 \right) \right] C_i$$

The computed rotor thrust for a given power is determined by iteration of the above equation. A numerical example of computation of rotor thrust is given in the next Section.

SECTION 6

RESULTS OF PERFORMANCE COMPUTATIONS

The analysis developed in this report was used to compute design rotor thrust for detail design of the Hot Cycle rotor, as required by Item 4e of the contract, Reference 1. The geometry (that is, radius, chord, thickness, twist and duct area) were determined on the preliminary design study reported in Reference 7. That report discussed a helicopter of 11,520 pounds gross weight, using two gas generator versions of the Lycoming T53 turboshaft engine, designated Lycoming B.

Shortly after the Reference 7 study was submitted, preliminary information was obtained on the characteristics of the General Electric T64 engine. That engine is a more sophisticated, higher compression ratio engine than the Lycoming B. The higher compression ratio T64 also discharged its gas at a considerably higher pressure ratio than the Lycoming B, at full power. However, the T64 gas temperature at full power was about 300°F higher than the Lycoming B discharge temperature. The idea was developed that the T64 could be used as a gas generator of higher pressure and higher power than the Lycoming B, provided the gas temperature was reduced to the Lycoming B value (about 900°F) for which adequate duct material was available.

It was found that the T64 gas temperature at full power could be reduced very easily by injecting water into the tailpipe of the engine during the takeoff condition. Cruise analysis showed that best cruise speed and speed for onset of rotor stall were both near 100 knots, a speed at which the two engine power required was low enough that the gas temperature to produce that power was actually less than 900°F. Therefore, using tailpipe water injection for takeoff only, it was possible to design the helicopter to take advantage of the power and high compression of the T64 to establish the gross weight. Although the T64's would be operated at part throttle in cruise, their fuel consumption was still better than the Lycoming B because of the better thermal cycle of the T64. (The water flow rate for takeoff only was found to be only 1% of gross weight for each three minute full power takeoff).

Using preliminary T64 gas conditions and assuming tailpipe water injection, the following design performance was computed.

a. Design Parameters of the Hot Cycle Rotor System

Blade radius	$R = 27.5$ feet
Blade chord	$c = 31.5$ inches
Number of blades	$b = 3$
Airfoil section	NACA 0018
Duct Utilization	$U = .465$
Blade twist	$\theta = -8^{\circ}$
Tip speed	$V_T = 700$ ft/sec
Hovering altitude, out of ground effect	6000 feet
Ambient air temperature	$95^{\circ}F$
Blade duct friction coefficient	$f = .004$
Blade duct hydraulic diameter	$D = 0.448$ feet
Nozzle velocity coefficient	$C_{v_e} = .955$
Static duct total pressure ratio	$r_4/r_3 = .96$
Number of gas generators	2
Type of gas generator	General Electric T64 (2650 SHP)

T64 gas conditions @ 6000' - 95° (Preliminary) with tailpipe water injection at military power)

Mass flow	W_g = 19.5 lb/sec
Pressure ratio	r_2 = 2.35
Temperature	T_{T_2} = 1360°R (900°F)

b. <u>Results of Performance Computations</u>

Blade duct inlet Mach number	M_3 = 0.39
Blade total pressure ratio	r_4/r_3 = 1.001
Tip nozzle total pressure ratio	r_4 = 2.255
Jet velocity	V_j = 1760 ft/sec
Rotor horsepower per lb per sec of gas	$\frac{RHP}{Wg}$ = 41.9
Rotor horsepower	2 x 19.5 x 41.9 = 1632
Rotor thrust	T = 15300 lbs

Subsequent to the determinations of the design rotor thrust of 15300 using T64 gas generator performance and tailpipe water injection, information became available on a newer material called Rene' 41 which could be used for the ducting, and which would permit operation of the gas generators at full temperature without using water injection. Performance computations based on use of full engine temperature as well as pressure indicated substantial improvement could be obtained in the important parameter of payload/empty weight, even though impressive values were already available. Negotiations were conducted, leading to the contract modification of Reference 8. That agreement provided for the use of the Rene' 41 alloy as duct material and was based on the use of the full temperature and pressure output of the T64 gas generator. Performance computations that were made based on full T64 output temperature, (plus revised and improved gas conditions supplied by General Electric instead of the earlier estimates made before the engine had actually run) indicated that 17200 pounds rotor thrust could be developed at 6000 feet - 95°F. Since detail design of the whirl tower hub structure had just about been completed, it was decided to retain the lower design figure of 15300 pounds at 2.5g load factor). For operation at higher gross weight at reduced load factor, a hover weight of 17200 pounds at 6000 - 95°F could be attained. The blades, of course, would be capable of operating with full temperature, following the change to Rene' 41 ducting material.

SECTION 7

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